Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 000:00:14:30.011	Problem	**FIAR**	**IFA** STS-54-V-01	HYD
MMACS-01	**GMT:** 013:14:14:00.000		**SPR** 54RF01	**UA**	**Manager:**
			IPR	**PR** HYD-5-04-0104	
					Engineer:

Title: WSB 3 No Cooling (ORB)

Summary: DISCUSSION: During Ascent, water spray boiler (WSB) 3, serial number (S/N) 15 exhibited no cooling until just after the early shutdown of auxiliary power unit (APU) 3. The APU 3 lube oil return temperature reached approximately 295?F when the WSB was switched from controller "A" to controller "B". The lube oil return temperature reached 315?F when the decision was made to shutdown APU 3 early (lube oil return temperature peaked at 319?F and APU bearing temperatures peaked at 340?F). After APU 3 deactivation, the WSB 3 GN2 regulator outlet pressure indicated spraying had started. WSB 3 continued to spray until the spray logic was turned off (total spray time was approximately 35 seconds). Steady-state cooling was never achieved on controller "A" or "B".

APU 3 was selected to perform the flight control system (FCS) checkout. The FCS checkout timeframe was extended to verify WSB 3 cooling performance. The extended APU 3 run-time demonstrated satisfactory cooling on both controllers, with a minor overcool observed on controller "A". APU 3 performance during entry on the "B" controller was nominal. WSB S/N 15 was installed on OV-104 prior to its first flight. The boiler was flown on two flights of OV-104 with no major cooling anomalies noted. In November 1986, the WSB was found to have extensive shell corrosion. The WSB was removed and returned to the vendor for cleanup and refurbishment, which included a new aluminum shell. The baseline acceptance test procedures (ATP) were performed on the WSB prior to shipment back to the field. WSB S/N 15 was installed with an Improved APU on OV-105 at Palmdale in late 1990 prior to its first flight. No cooling anomalies were observed with this WSB during the first flight (STS-49) of OV-105. During STS-47, the following flight, the WSB lube oil spray bar froze on ascent with cooling occurring just prior to an early APU shutdown. The return temperature climbed to 311?F at APU shutdown. After STS-47, the WSB was hot oil flushed as a result of this anomaly. Due to the subsequent ascent cooling failure seen during STS-54, it must be concluded that the hot oil flush had no effect. Since the spray valve and both controllers were verified to be operating satifactorily during FCS checkout and entry (controller "B") of STS-47 and STS-54, spraybar freeze-up remains the most probable cause of this cooling problem. Data analysis also indicates that the local pressure during ascent at the vent nozzle of system 3 is somewhat higher than the other two systems. The high pressure is due to the location of the system 3 vent nozzle outlet (it is farther forward than the system 1 and 2 vent nozzle outlets). System 3's pressure remains higher than the other system for the first 80 seconds of ascent which is believed to be a contributing factor toward the repeated freeze-up anomalies observed in system 3 of all vehicles. Boiler performance during

FCS checkout on "A" and "B" controller and entry performance on "B" controller appeared nominal. However, an"A" controller electrical checkout and spray sensor "A" checkout were performed due to the minor overcool seen during FCS checkout while on "A". No anomalies were noted during checkout. Spraybar freeze-up conditions occur when the water triple point condition is reached inside the heat enchanger. In the worst case freeze-ups, it is postulated the water triple point was reached prior to MECO. By inceasing the water preload, the duration of heat exchanger tube bundle/water preload contact can be increased which will reduce the likelihood/severity of spraybar freeze-up by maintaining the pressure above the water triple point past MECO. The ongoing spraybar freeze-up test analysis indicates that the severity of the spraybar freeze-up at water triple point conditions may inversely correlate to the amount of water in the boiler. Therefore, KSC, as requested, has increased the water preload of WSB 3 to 5.0 +/- 0.1 lb water (the current nominal water load is 3.75 +/- 0.25 lb) on STS-57. CONCLUSION: The failure of the WSB to adequately regulate the lube oil temperature during ascent resulted from a freeze-up of the APU lubrication oil water spraybar. CORRECTIVE_ACTION: The next flight of OV-105, the WSB (3) will be flown with an increase in the water preload to 5 lb. EFFECTS_ON_SUBSEQUENT_MISSIONS: Possible lubrication oil spraybar freeze-up ascent.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR** BFCE 029F066,	**IFA** STS-54-V-02	GFE
INCO-01	**GMT:**		029F064, 029F065,	**UA**	**Manager:**
			029F067	**PR**	
			SPR		**Engineer:**
			IPR DR BH330019,		
			BH330018, BH330055,		
			BH330054		

Title: Camera AnomaliesA. CCTV Camera D No Image (Intermittent)B. CCTV Camera B Problem During Split Screen OperationsC. CCTV Camera A Momentary Red and Green LinesD. CCTV Camera C High Gain Anomaly (GFE)

Summary: DISCUSSION: A. On flight day 1, closed circuit television (CCTV) camera D originally produced normal, good quality video of payload bay scenes. Approximately 5 hours into the flight, it was discovered that the camera temperature had reached 54?C after the camera had remained in operation for an extended period of time. The 54?C temperature is 9 degrees above the redline temperature at which the caution and warning system should sound an alarm. However, the alarm had been disabled to permit use of the camcorder on the downlink. Although the camera should not have experienced a failure solely due to reaching 54?C, the high temperature most likely contributed to the failure. CCTV camera D was checked later in the mission and the quality of the video was intermittently normal. The camera was removed from the vehicle and sent to the vendor.

B. During the extravehicular activity (EVA) on flight day 5, CCTV camera B was being used in the split-screen mode and a synchronization problem occurred between camera B and the CCTV system. After the camera continued to operate nominally. Analysis indicated that the problem was caused by camera B and not the video switching unit. (The VSU sets up the split screen. The picture on camera B was there, but the bottom half of the picture was in the center. This occurred on both sides of the screen.) The camera was removed from the vehicle and returned to JSC for evaluation. The camera was tested and operations were nominal. It will be returned to inventory and will be available for flight usage. C. On several of the separate downlinks of CCTV A video, momentary red and green horizontal lines were noted near the lower quarter of the image. The camera was removed from the vehicle and returned to JSC for evaluation. The camera was tested and operations were nominal. This camera will be used only as a last resort. D. When looking at a low-light scene with CCTV camera C, the entire scene was not shown (i.e., only the brightest stars were noted). There should have been noise in the picture, but there was none. The camera was removed from the vehicle and sent to the vendor. CONCLUSION: A. A most probable failure for camera D is unknown at this time. B. Most probable cause is a failure in the camera B synchronization not recognizing the boundary lines for the split-screen sides. This occurs when the camera is shut off and powered-on quickly, causing a memory reset and locking up the split-screen system. C. Most probable cause is a degradation in the camera's high-voltage supply. D. Most probable cause is a failure in the high-gain circuitry. CORRECTIVE_ACTION: A. The camera has been sent to the vendor for failure analysis. Any further troubleshooting and results will be tracked by FIAR BFCE 029F066. B. The camera synchronization for split-screen operations can be corrected by cycling camera power. This camera will be placed in inventory. C. The conditions noted on this camera are the indications of high voltage power supply degradation. High-voltage power supplies are no longer available. Most power supply degradations are seen in zero-gravity vacuum conditions. Using this camera in a Spacelab or in the cabin may prolong the life of this power supply. D. The camera has been sent to the vendor for failure analysis. Any further troubleshooting and results will be tracked by FIAR BFCE 029F067. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 003:00:22:39.011	Problem	**FIAR**	**IFA** STS-54-V-03	ECLSS
EECOM-04	**GMT:** 016:14:22:09.000		**SPR** 54RF02	**UA**	**Manager:**
			IPR 57V-0006	**PR**	
					Engineer:

Title: EDO WCS Commode Fault Light On (GFE)

Summary: DISCUSSION: At approximately 016:14:22:09 G.m.t. (03:00:22:39 MET), the Extended Duration Orbiter (EDO) waste collection system (WCS) commode fault light illuminated. A review of data indicated that the compactor caused the fault light to illuminate during the retraction phase of the compaction cycle. An in-flight procedure was performed and confirmed the fault. There were two most likely causes of the fault. First, the compactor piston may not have been returning to the fully-nested position in the housing, resulting in the piston not activating the top-of-travel limit switch. However, a flight day 5 test performed by the crew verified proper operation of the limit switch. The other most likely cause of the problem was that the controller logic may have been sensing the current-limit condition too quickly at the end of the cycle, causing the overcurrent sensor to shut the compactor down. The crew was told to cycle the power on the commode prior to use if the light was on, as this reenabled the motor. The WCS operated satisfactorily for the remainder of the mission.

Postflight testing, performed by vendor personnel did not duplicate the problem; however, it revealed that the foam snubber that was installed to limit the current-rise rate was stiffer and visually smaller. When the snubber was initailly installed, it introduced a 300-msec delay into the controller logic that was sensing the current-limit condition. When tested postflight, this delay had degraded to 200-msec. CONCLUSION: Degradation of the snubber allowed a current spike to be produced that the controller interpreted as an overcurrent condition and shut the system down. CORRECTIVE_ACTION: Fabricate and install a new snubber to support STS-57. Train the crew to recognize this failure and provide updated procedures onboard to continue operations, if this condition were to occur on STS-57. Investigate modifications to the controller to lengthen the time between the current-limit condition and the overcurrent condition, thus eliminating the need for the snubber. This will be considered for implementation to the third flight of the EDO WCS. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 003:07:06:30.011	Problem	**FIAR**	**IFA** STS-54-V-04	HYD
MMACS-03	**GMT:** 016:21:06:00.000		**SPR** None	**UA**	**Manager:**
			IPR 57V-0012	**PR**	
					Engineer:

Title: Rudder Speedbrake Switching Valve Indication (ORB)

Summary: DISCUSSION: During the on-orbit phase of the STS-54 mission, the rudder/speed brake (RSB) switching valve position indication (V58X1001E) did not indicate a switch as expected to its second standby position when hydraulic system 3 circulation pump was turned on. Pressure at the circulation pump reached ~300 psi with a temperature at the speedbrake of ~10?F. Also, it did not indicate a switch back to the system 1 (primary) when the circulation pump was turned off.

An on-orbit troubleshooting plan was developed to test the switching valve for movement at a higher pressure. It was decided to wait until the hydraulic system 3 circulation pump would be required for thermal conditioning of the rudder/speed brake hydraulic return line. The line reached -10?F. and the circulation pump was turned on (~017:16:16:20 G.m.t.). The thrust vector controller isolation valve was commanded open (~17:16:19:08 G.M.T.) by the crew to increase the system pressure. The result was ~475 psia with the switching valve moving to its second standby (~17:16:19:09 G.m.t.). The failure of the RSB switching valve to indicate a change to the system 3 (second-standby) is due to a below specification amount of pressure at the valve to cause the valve to switch positions. Procurement specifications require the switching valve to switch to its second standby system at 200 psi when all other ports are at 50 psi. The pressure measurement at the circulation pump is not an accurate representation of the pressure at the switching valve. This is due to the distribution of the circulation pump output to many other actuators and line losses. Line losses are much more pronounced at very low temperatures. Failure of the switching valve position indication to indicate a change back to its hydraulic system 1(primary) position after system 3 hydraulic circulation pump was turned off is the result of differences in reservoir pressures between hydraulic systems 1(primary) and 3(second-standby), and the temperature induced viscous friction forces between the spool and sleeve. Hydraulic system 3(second-standby) reservoir pressure was higher than hydraulic system 1(primary) reservoir pressure. This caused the switching valve to try to remain in the second-standby position and oppose the force of the spring which us trying to return the valve to its system 1(primary) position. The spring force on the spool to return it to its system 1 (primary) position is 22 lb. The high pressure in system 3(second standby) coupled with the friction force between the spool and the sleeve, prevented the spring from returning the switching valve spool to its system 1 (primary) position.

The viscous friction force analysis indicated that at -10?F, the force required to move the switching valve spool could be as high as ~77 lb and at 10?F the force could be as high as ~45 lbs. The friction force at these different temperature are all much higher than the springs capability. However, there is currently no requirement for switching valve performance when circulation pumps are turned off. Postflight troubleshooting with the hydraulic ground test carts and circulation pumps indicated nominal performance of the RSB switching valve. The RSB switching valve performance, as experienced on this flight, should be expected on future flight of all vehicles. CONCLUSION: The switching valve will switch positions when the minimum specification pressure differential is met. The circulation pump pressure measurement is not a good indication of the pressures at the switching valve. At very low temperatures and pressures, the switching valve spring may not return the spool to its system 1 (primary) position.   CORRECTIVE_ACTION: The Shuttle Operational Data Book is being updated to reflect the switching valve performance at low temperatures and pressures.   EFFECTS_ON_SUBSEQUENT_MISSIONS: None

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Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:** 004:17:58:30.011	Problem	**FIAR**	**IFA** STS-54-V-05	RCS
PROP-01	**GMT:** 018:07:58:00.000		**SPR** 54rf03	**UA**	**Manager:**
			IPR	**PR** RP04-11-0353	
					Engineer:

Title: R1R Failed-Off (ORB)

Summary: DISCUSSION: Reaction control subsystem (RCS) primary thruster R1R was declared failed-off during the RCS hotfire test. This was the first attempted firing of thruster R1R during the mission. When the fire command was initiated, the thruster chamber pressure increased to 10 psia for the first 280 msec of the firing and then stepped up to 25 psia prior to deselection by redundancy management (RM) at 320 msec. RM declares a thruster failed-off after receiving three consecutive chamber pressure discretes indicating a chamber pressure of less than 36 psia. The nominal chamber pressure for a primary thruster is 152 psia.

Injector tube temperature data indicates both fuel and oxidizer flow. The oxidizer flow was most probably pilot-valve-only flow, which accounted for the low chamber pressure. The oxidizer valve main stage probably failed to open due to iron nitrate contamination of the pilot stage. The increase in chamber pressure just prior to thruster deselection could be indicative of a sudden increase in oxidizer pilot flow with a possible partial opening of the main stage. The thruster was left deselected for the remainder of the mission.   Thruster R1R (S/N 137) has flown 12 missions, 6 with the current oxidizer valve. The oxidizer valve is a -506 configuration and the thruster will be sent to the White Sands Test Facility (WSTF) for valve flushing. If flushing is successful, the thruster will be returned to service. If flushing fails, the thruster will be returned to the vendor for failure analysis.   CONCLUSION: The most probable cause of the thruster fail-off was iron nitrate contamination in the oxidizer valve pilot stage that prevented its proper operation.   CORRECTIVE_ACTION: KSC has removed and replaced thruster R1R and it will be transfered to the WSTF for the thruster flush programs. Iron nitrate formation is assisted by the presence of water (moisture) in the oxidizer valve. Therefore, the primary thruster throat plugs are installed during turnaround to reduce the likelihood of moisture intrusion into the propellant system. Results of the thruster flush at the WSTF and any necessary failure analysis will be documented in CAR 54RF03.   EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	MET:	Explained Condition	FIAR	IFA STS-54-V-07	FCP
EGIL-02	GMT: 019:13:30		SPR 54RF05	UA	Manager:
			IPR None.	PR	
					Engineer:

Title: Fuel Cell 2 and 3 Alternate Water Lines Temperatures Increasing (ORB)

Summary: DISCUSSION: During on-orbit operations, the temperature of the fuel cell 2 alternate water line was higher than normal, indicating water weeping past the fuel cell 2 alternate water line check valve. During entry at 19:13:30 G.m.t. (05:23:31 MET), the temperature of the fuel cell 3 alternate water line began rising and peaked at 127 degrees F approximately 30 minutes after landing. The alternate water line temperature peaked at the same temperature as the primary water line, indicating flow past the fuel cell 3 alternate water line check valve.

After the mission, the alternate water line relief valves were tested for proper cracking and reseat pressures. The fuel cell 2 valve was found to crack at a differential pressure of 6.05 psi (specification = 5.5 psi minimum), reseat at 5.05 psi (specification = 5.0 psi minimum), and its post-reseat leak rate was measured to be 13.5 ml per 3 minutes (specification = 15 maximum). The fuel cell 3 valve cracked and reseated exactly at the specification values. CONCLUSION: Both the fuel cell 2 and 3 alternate water line relief valves were within specification, however, both were very close to the limits. By being so close to the limits, these valves allowed small amounts of water to flow into the alternate water lines, causing the temperature profile noted. CORRECTIVE_ACTION: None. Fly as-is. These valves are within OMRSD specification. The small amount of water introduced into the alternate water line is inconsequential. Flow through the alternate water lines becomes a flight impact only if caused by constriction of flow in the primary water path, which was not a factor in this case. EFFECTS_ON_SUBSEQUENT_MISSIONS: None. Expect similar alternate water line thermal traces as described above.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	MET: 000:00:14:30.011	Problem	FIAR	IFA STS-54-V-08	HYD
MMACS-02	GMT: 013:14:14:00.000		SPR 54RF09	UA	Manager:
			IPR 57V-0008	PR	
					Engineer:

Title: Hydraulic system 3 pump outlet pressure high post-ascent APU shutdown (ORB)

Summary: DISCUSSION: Hydraulic supply pressures in system 3 and later system 1, showed an anomalous pressure recovery immediately following the post-MECO APU shutdown. The system 3 pressure, following an initial drop from 3000 psia to 1600 psia over 6 seconds, increased over the next 4 seconds to ~2400 psia and remained there for more than 40 seconds. During the next 8 seconds, when the TVC isolation valves 1, 2, and 3 were closed in sequence, system 3 pressure again dropped to the 1600 psia range, sharply recovered to 2800 psia and finally dropped to the expected reservoir pressure. After APU 1 shutdown, hydraulic system 1 also showed an unexpected transient pressure recovery.

Initial trouble-shooting focused on intersystem leakage at the switching valves as the method for transferring power into the shutdown systems. However, the overall flight data did not support this mechanism of energy transfer. Continuing analysis subsequently determined that the pressure recovery in system 3 was instead caused by backdriving the no. 3 speed brake hydraulic motor, converting the motor into a hydraulic "pump" and re-pressurizing the shutdown system. Power to operate the "pump" was delivered through the differential gearbox that normally combines the outputs of the three speed brake motors. APU's 1 and 2 showed an abrupt 8 horsepower increase during the 40 second period, which closely correlates with the calculated 8.4 horsepower required in system 3 to pump the estimated 6 GPM quiescent flow required to maintain the observed 2400 psia. During the 40 second period when back driving of the system 3 speed brake motor occurred, the rudder servo driver currents and differential pressures showed changes with the onset of the pressure recovery. The rudder driver currents increased ~0.31 m amp for an ~8 second period and rudder delta pressures increased ~125 psid for ~9 seconds. These changes indicate that the motors were responding to changes in the rudder positions. These same parameters showed similar signatures during nominal APU 1 shutdown and resulting hydraulic system 1 supply pressure recovery. As a comparison, these paramters do not show a shift during normal APU shutdown where pressure recovery does not occur. An integral part of each motor is a spring-loaded locking brake. Its design function is to lockup a failed or shutdown motor to prevent a functioning motor output torque from being lost by back-spinning through the differential gearbox. The brake uses high system pressure (>2200 psi +/- 100 psi) to hold the brake off, and upon loss of pressure, the system relies on the compressed springs to apply the brake as supply pressure bleeds downs. The procurement specification specifies: the motor/brake completely releases from 2400 to 3100 psi, to be capable of developing the required holding capacity to react to the maximum limit hinge moment from 0 to 1000 psi, and from 1000 to 2400 psi the combined motor brake holding capability shall be equal to or greater than 2.814 million in-lb hinge moment for the speedbrake, and 0.92 million in-lb hinge moment for the rudder. The vendor's Acceptance Test Procedure(ATP) demonstrated the motor/brake performance by: operating all three power drive unit motors, 2 motors, and each motor against the operating hinge moment. The vendor tests the brake component by verify greater than 540 in-lb of holding torque is available at zero pressure and the brake will release at 2200 +/- 100 psi (turns freely with 2.2 in-lb torque). However, the brake relock performance with a decreasing pressure was not specifically demonstrated. Two concurrent conditions must exist before backdrive can occur: the speed brake must be commanded to stall torque conditions and supply pressure in one of the hydraulic supplies must have declined to a threshold range to create a differential torque between adjacent motors. During ascent, and throughout entry until Mach 10, a -9 degree position is imposed on the speed brake. This causes each of the 3 speed brake motors to deliver their full stall-torque into the differential gearbox. The torque output of a given motor at stall conditions is a direct function of that specific supply pressure. When pressure decreases within a system, output torque also decreases causing torque unbalance to increase between adjacent motors. At some point, the torque unbalance becomes sufficient to overcome the breakaway torque (static friction) that tends to hold the motor/gearbox assembly stationary. Once "breakaway" occurs, the abruptly lowered dynamic friction allows the differential torque to quickly backdrive the reduced pressure motor, causing it to become a hydraulic pump and increasing the pressure in the low pressure system. The role of static motor breakaway friction is a major factor in determining if backdrive can occur. Discussions with the motor manufacturer indicate static motor breakaway torque is not always predictable and is subject to abrupt changes. The assessment is supported by previous flight experience. A review of all flight data revealed 37 cases of concurrent hard-over commands and adeqaute time/differential-pressure between system pressures during post-MECO APU shutdowns. These 37 flight cases inlcuded at least one instance where each individual motor/brake assembly within the fleet was exposed to the backdrive conditions, yet successfully allowed normal shutdown. Only one additional case of motor backdriving and pressure recovery was found; STS-36 system 1 during early entry. In that case, the system failed to properly depressurize when commanded, instead abruptly recovered at 2100 psi, then stabilizing in the 2500 psi range for ~7 minutes. Occasional bodyflap commands caused 250 psi transients, plus other flow demands caused various drops to the 1700 psi range. The last transient apparently dropped the pressure sufficiently to engage the locking brake and allowed the system pressure to finally drop to the depressurized value. At that time, the problem was thought to be caused by a sticking main pump compensator. Tear down inspection revealed evidence of contamination related damage and witness marks in the compensator area, supporting the assumption of a transient contamination related problem. However, STS-36 data showed similar signatures in the rudder driver servo currents and delta pressures. These data suggest the same hydraulic supply pressure recovery associated with back-driving. To gain insight into the brake performance for a decreasing pressure condition, a series of engineering evaluation tests were recently completed on the 6 motor brakes from the qualification test unit. The results indicated that the locking-brake remains completely off until presure decreases to the 1800 psi range. As pressure continued to decrease further, brake torque showed the expected linear increase with decreasing pressure, uniformly increasing to the 800 in-lb range as pressure approached zero. These data were repeatable among the 6 units. These results indicate that the static holding torques of the stationary motors is the only factor that prevents motor backdrive until pressure declines to the 1800 psi range. This, along with flight experience, indicate the static breakaway friction normally is above the breakaway torque threshold; this allows hydraulic pressure in the shutdown system to continue decreasing, quickly achieving sufficient brake lockup to prevent motor backdrive. Since, the three in-flight cases of backdrive all occurred during relatively inactive periods of flight, a detailed assessment was performed to determine the flight control performance should backdriving occur during the more critical fight phases. In an attempt to "bracket" the performance question, mission profiles were analyzed for all phases of ascent and entry, including abort cases. These mission models were run assuming a supply pressure of only 1600 psi. This value was selected based on two pertinent parameters: This is the pressure range where switching valves will de-select a failed systems, which coincidentally, is also the threshold range where the sum of the locking brake torque and backdrive motor torque will avoid backdriving. Results of this study identified several important system design functionality points: First, a hydraulic supply pressure limited to 1600 psi can meet all flight control hinge-moment requirements. This indicates that the actuators will not stall, but rather will move against their opposing air loads. Second, flight experience as well as analysis indicates the reduced pressure condition would be a transient condition. The maximum pumping capacity of a backdriving motor is ~10 GPM, whereas required transient flow demands can be in the 42 GPM range. Transient flow demands in active phases of flight will "over-demand" the backdriving "pump". This has the positive effect of temporarily dropping the supply pressures to the level where the switching valve will switch the standby system to attempt to meet the high flow demand. Additionally, pressure transients should cause the motor locking brake to engage, locking up the backdriving motor, and allowing the shutdown system to finally bleed-down. As observed in both STS-36 and STS-54, transient over-demand will sharply drop system pressure, causing switching valves to select the respective standby systems. This indicates that a backdrive condition will eventually "cure" itself. This assessment also determined that backdrive can occur only when the speed brake has a hard-closed command. Once the speed brake is opened, the system pressure required to develop motor torque needed to meet aerosurface hinge-moments is below the threshold value of the motor backdrive torque. Peak motor output torques required to meet rudder and speedbrake aerodynamic requirements are ~30 in-lb to ~130 in-lb, compared to a peak running torque of 245 in-lb. The differential pressure required to produce 130 in-lb torque would at best produce only about 1400 psi in the degrading system, which is below the conditions where the locking brake will allow backdriving. The net effect is that backdriving is not an issue for entry once the speed brake is opened. It should also be noted that the STS-36 incident occurred before Mach 10, while the speed brake was still commanded to a hard-closed position. This review has shown that sufficient data are available to assure that the flight hardware will meet its intended function. This confidence is because the flight control system, rudder, and speedbrake, have substantial design margins. Breakaway torque was measured during acceptance test for each brake module subassembly for a zero-pressure condition. This verified that the locking brake breakaway torque always exceeded the minimum allowable of 540 in-lb; with the typical value in the 800 in-lb range. That same test series also verified that the individual brakes would hold a 25 in-lb torque until at least 2100 psi, and as pressure continued to increase, holding torque was less than 2.2 in-lb at 2700 psi. (motor stall torque 208 in-lb min.) Additionally, the engineering measurements determined locking brake breakaway torque at decreasing pressures all followed a repeatable holding torque vs. pressure profile. Combining these three sets of data provide rationale to conclude that the locking brakes on both the rudder motors as well as speed brake motors should lockup and prevent a possible backdrive condition, once the speed brake is opened. This assessment was based on taking the brake module lockup torque pressure/lockup torque tests. The data was then correlated with the torque/pressure curve for the rudder and the speed brake hinge-moment requiremens. This does provide the rationale to conclude the hardware will meet the intended function, even though it was not verified in acceptance test for the declining pressure condition. CONCLUSION: The conditions leading to the hydraulic system supply pressure recovery were caused by backdriving of the speedbrake motor. CORRECTIVE_ACTION: Short-term - None, fly-as-is. Analysis, test data, and flight data have shown that the flight system can accommodate a repeat of this phenomena with no impact to flight safety. Long-term - A test program is being developed to better quantify the static/dynamic breakaway friction of the hydraulic motor/brake. Resolution: Closed. EFFECTS_ON_SUBSEQUENT_MISSIONS: Reoccurrnce of supply pressure recovery due to backdriving phenomenon.

Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-54-V-09	D&C
EGIL-01	**GMT:**		**SPR** (A) 54RF-06, (B) 54RF07, (C) 54RF08	**UA**	**Manager:**
			IPR 57V-0010	**PR** (A) DDC-5-04-0040, (B) -0041, (C) 0042, (FEA) FEL-	**Engineer:**

Title: Floodlight FailuresA. Forward StarboardB. Mid StarboardC. Mid Port (ORB)

Summary: DISCUSSION: During the payload bay floodlight activation on flight day 4, the forward starboard and mid starboard floodlights did not come on. The mid port floodflight came on approximately one hour after switch activation.

A. Analysis of the forward starboard floodlight data indicates that the associated remote power controller (RPC) had tripped, which suggested arcing had occurred in the floodlight assembly. The KSC visual inspection verified arcing in this light. B. Analysis of the mid starboard floodlight data indicates that the RPC had tripped which suggested arcing had occurred in the floodlight assembly. The KSC visual inspection verified arcing in this light. C. The mid port floodlight data indicates that the lamp did draw current, but failed to illuminate in the expected time (5 minutes). The KSC inspection indicated arcing in this light. The arcing problems which are causing the floodlights to fail in-orbit are related to the loss of the nitrogen backfill in the floodlights. The lights are sealed at 0.5 atm with pure nitrogen at the NASA Shuttle Logistics Depot (NSLD), but with time this backfill decays to the point where Corona effects begin to occur due to the high voltge required to ignite the floodlights. Rockwell is currently analyzing the seal design and is expected to provide design change recommendations to preclude floodlight backfill leakage. In addition, a fix for the floodlight electronics assembly (FEA) has been approved that will provide a greater starting potential to the lights and also correct a duty-cycle problem. However, there are not yet any modified FEA's in the fleet due to a backlog of unrepaired floodlight hardware at the NSLD. CONCLUSION: A & B. Loss of the backfill on the forward starboard and mid starboard floodlights caused the arcing conditions on the lights and led to overstress of the FEA because of the high current flow. C. The most likely cause of this failure is the marginal starting potential available from the FEA to ignite the light coupled with arcing due to backfill loss. CORRECTIVE_ACTION: The three floodlights and FEA 2 which drives the forward starboard and mid starboard lights were removed and sent to NSLD for troubleshooting. The troubleshooting results and any further correctve actions will be tracked under the noted CAR's. EFFECTS_ON_SUBSEQUENT_MISSIONS: Random floodlight failures may continue to occur until all hardware has been modified to the new configuration. The number of floodlights required is mission dependent. There are six payload bay floodlights and one bulkhead floodlight, that are each driven by independent ballasts within the FEA. Generally, there is sufficient redundancy to provide adequate lighting for isolated failures.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 005:23:27:30.011	Problem	**FIAR**	**IFA** STS-54-V-10	APU
MMACS-04	**GMT:** 019:13:27:00.000		**SPR** 54RF12	**UA**	**Manager:**
			IPR 57V-0007	**PR**	
					Engineer:

Title: APU 3 Bearing Temperature Erratic (ORB)

Summary: DISCUSSION: During STS-54 entry, approximately 35 minutes after APU start, the APU 3 gearbox bearing temperature 2 (V46T0362A) became erratic for a period of about 17 seconds. It then recoverd and operated nominally throughout the remainder of the mission and during postlanding soakback.

Postflight troubleshooting included a sensor wire wiggle test and demate and inspection of connectors J3 at the APU and 54P66 at dedicated signal conditioner OA1. The problem could not be repeated. CONCLUSION: The most probable cause of this anomaly is an intermittent open circuit in the temperature sensor signal path. CORRECTIVE_ACTION: The bearing temperature sensor has been removed and replaced. This includes the wiring from the sensor to the APU connector. An unexplained anomaly (UA) is currently being processed. EFFECTS_ON_SUBSEQUENT_MISSIONS: None. The APU 3 bearing temperature sensor 1 (V46T0361A) can be used as a backup. If this sensor also fails, the lube oil outlet and return temperature sensors can be used to evaluate the health of the APU and lube oil system.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-54-V-11	EPD&C
None	**GMT:**		**SPR** 54RF10	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: Extravehicular Mobility Unit Power Supply and Battery Charger Noise ()

Summary: DISCUSSION: The crew reported that the extravehicular mobility unit (EMU) power supply and battery charger (PS&BC) exhibited excessive audible noise. This noise was only present when the unit was charging the EMU. The EMU PS&BC provides power to the airlock for the startup of the EMU in preparation for an extravehicular activity (EVA). In addition, the unit also provides a constant current for charging the batteries in the suit. The unit consists of two channels having a power supply and a battery charger on each channel. Since the charger and power supply circuits of one channel are common, only one function can be performed at a time by each channel. However, the two channels can perform separate operations (one channel can be in the power supply mode and the other channel can be in the battery charger mode). The PS&BC is mounted on a coldplate in the forward avionics bay 1A.

The EMU PS&BC will not be removed from the vehicle at this time. Based on laboratory tests, the available spare may be noisier than the unit onboard OV-105.
CONCLUSION: The most probable cause of the noise is within the PS&BC inverter; however, the actual cause needs to be isolated. CORRECTIVE_ACTION: Current plans are to measure the acoustical-noise level on as many Orbiters as possible to determine Orbiter vehicle end item (OVEI) specification compliance, acoustical impact, and the potential need for corrective action to the PS&BC. The addition of such measurements is being considered for STS-55 (OV-102) and STS-57 (OV-105) so that corrective measures will be in place for STS-61 (OV-105). STS-61 is the Hubble Space Telescope revisit mission and will have continuous PS&BC use for up to 5 days because of consecutive EVA's. The gathered data, conclusive results, and any further corrective actions will be tracked by CAR 54RF10.
EFFECTS_ON_SUBSEQUENT_MISSIONS: Some STS-57 crew members may have difficulty sleeping because of the PS&BC noise. Once the data are gathered on STS-57 and evaluated, the appropriate corrective measures will be taken to correct the noise problem.
